HIGH ALTITUDE SOUNDING ON MARS WITH AN INFLATABLE HYPERSONIC DRAG BODY (BALLUTE)

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ABSTRACT

Presented in this paper is a concept to probe the tenuous upper atmosphere of planet Mars by means of a hypersonic drag balloon, a device known as a “ballute”. This concept has been under active development for the past seven years by the Mars Society Germany and the University of the Federal Armed Forces of Germany in Munich, with further support by the AMSAT-DL .e.V. organization, the DLR and several other research institutions and industrial companies.
In addition to this application, the concept was also studied for applications in other fields, specifically emergency low Earth orbit recovery and delivering payloads to high altitude landing sites on Mars, such as Tharsis.
In this paper, the theory behind such a mission will be discussed, along with experience gained during its practical implementation.

1 INTRODUCTION

Any concept of flying on Mars , be it with static or dynamic lift, is restricted to altitudes well below 10 km. Since any sustained flight at that altitude or higher in such a thin atmosphere is technically impractical, so is ascending to that altitude with a sounding rocket. Yet providing an instrument suite with access to a large altitude range and a meaningful dwell time is of high scientific interest. A Scientific Committee, formed by scientists from several research institutes around Europe, drafted a reference scientific mission scenario [1] which was used by engineers and scientists from the Mars Society Germany (MSD) and the University of the Federal Armed Forces of Germany in Munich (UBW) as a reference baseline in a study to find a practical technical solution.

1.1 Engineering Background

Ascending from the surface of Mars with a balloon or a sounding rocket is difficult at best, not the least because relatively high wind speeds thwart the deployment of large gossamer balloon hulls or aeroplane wings without considerable mechanical effort [2]. Instead of ascending from the surface however, a high altitude mission might also descent from space, provided it can be built to decelerate at a sufficiently high altitude to make scientific gains desirable. No conventional hypersonic aeroshell and parachute system will slow down sufficiently to allow deployment far enough above the surface, because its mass to drag ratio (expressed in the so-called “ballistic coefficient”) is much too high.
In summary, one can state that the effect of a lower ballistic coefficient raises the entire deceleration profile to a regime where the atmosphere is less dense (a higher altitude), prolongs descent time and lowers the aerothermodynamic heating load that the vehicle will have to absorb. This translates into an increased dwell time during descent, an increase in the accessible altitude range for in-situ measurements, and facilitates landing at a site with too high of an altitude for heavier and speedier probes (such as Tharsis on Mars). In addition to this, a low or even adjustable ballistic coefficient may also be useful during aerobreaking phases.
In order to achieve the highest possible altitude profile, the ballistic coefficient has to be as low as technically practical. This is achieved by using a large and light inflatable sphere, which in itself offers other benefits as well. Since it can be densely packed and deployed with a small number of moving parts, it offers an attractive alternative to rigid structures. So the same technology might also offer a practical alternative if a large aerodynamic heat shield for a heavy payload is needed, especially when a rigid heat shield would be too big for the biggest rocket payload fairing. Last but not least, the ballute technology might also offer an inflatable solar sail structure

1.2 State Of the Art

A very good overview of past concepts, actual attempts and future plans for planetary areal reconnaissance missions can be found in reference [3].
As of today, the VEGA Venus balloons remain the only planetary balloon mission ever realized [3]. Part of the reason is that flying on Venus is comparatively easy with the thick atmosphere the planet is covered with.

Mars, in contrast, poses a more complex challenge. At a mere 7 hPa average surface pressure, it is certainly thick enough to necessitate a hypersonic entry vehicle and a supersonic parachute system, but unfortunately not thick enough to use these for gently descending to the surface. So airbags or retrorockets have to be used for a soft touch down and any extended dwell time in the Martian atmosphere becomes a technical challenge that hasn't yet been met. It takes just six minutes for an ordinary lander to reach the surface. High altitudes on Mars have therefore remained inaccessible to instrument carriers so far.

The concept of using a balloon to alter the aerodynamic properties of a space vehicle for atmosphere manoeuvres was most prominently described in 1982 by Arthur C. Clarke in his novel "2010: Odyssey two". The original idea for an inflatable decelerator however is much older and was first investigated by the Goodyear Corporation in 1959 [4] (see Fig. 1).

Goodyear also coined term "ballute". The device was an almost spherical object that really had a notable resemblance to a balloon, intended to replace a conventional supersonic disc-gap-band style parachute in regions where either the speed of the vehicle or its altitude were considered too high for the reliable deployment and effectiveness of the latter [4][5]. One such application was the recovery system on the high-altitude ejection seats of the Gemini spacecraft, which used a ballute of Goodyear's design to recover astronauts forced to eject at great altitudes.

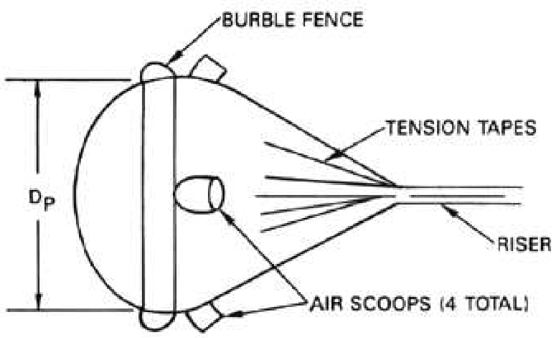


Fig. 1: Sketch of the Goodyear Ballute configuration

Today, the term "ballute" is used for any inflatable device intended to raise an object's aerodynamic drag, no matter the shape or intended purpose. The inflatable re-entry and descent technology demonstrator (IRDT) flown into orbit and back on February 9th, 2000 was the first atmospheric re-entry from space using an inflatable decelerator [6]. A consortium of Lavochkin and Astrium Bremen cooperated on the project after Lavochkin's earlier attempt at flying a similar entry vehicle on Mars-96, which perished on route to Mars.

IRDT was tested a total of two times and, although neither test performed as intended, a lot of experience could be gained. IRDT lowered the ballistic coefficient by one order of magnitude to somewhere on the order of 10^0 as compared to classic entry vehicles (on the order of 10^1).

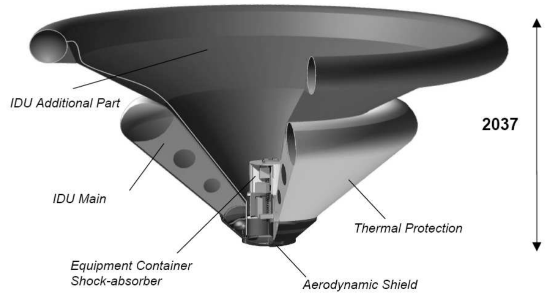


Fig. 2: Conceptual design of IRDT-2. [6]

The IRDT test vehicle is shown in Fig. 2. It had an equipment container and the inflatable decelerator referred to as IDU (Inflatable Deceleration Unit). The IDU was made of inflatable radial toruses that were interconnected by fabric sheets. The IDU could be inflated in two stages: the first stage (main stage) acted as an aerodynamic heat shield, while the second stage increased dynamic drag after peak heating, replacing a classic parachute system.

Today, studies of ballute applications exist aplenty, however few were ever realized. Studied ballute shapes do not only include spherical objects akin to normal balloons, but also toroids, cones and lentils [7][8][9]. Studied applications are mostly inflatable drag bodies intended to replace rocket engines for orbital manoeuvring, promising large savings in vehicle weight.

2 BASIC CONSIDERATIONS ON VEHICLES WITH LOW BALLISTIC COEFFICIENTS

2.1 Basic Theory

The ballistic trajectory of any object passing through the atmosphere of a planet is governed mainly by its velocity vector and its ballistic coefficient when it penetrates the sensible atmosphere. The ballistic coefficient sets into relation the mass of the object and the parameters governing its aerodynamic deceleration [10]. The ballistic coefficient is defined as

$$\beta = \frac{m_{sc}}{c_d A_{sc}} \quad \text{expressed in} \quad \left[\frac{kg}{m^2}\right] \tag{1}$$

where m_{sc} is the spacecraft's atmospheric entry mass, A_{sc} the drag effective face area normal to the velocity vector and c_d is the vehicle's coefficient of drag.

Since the aerodynamic drag on a body is given by

$$F_d = q_\infty c_d A_{sc} \quad ; \qquad q_\infty = \frac{1}{2} \rho_\infty |\vec{v}_\infty|^2 \tag{2}$$

with q_∞ as the free stream dynamic pressure and ρ_∞ as the free stream density [10][11], it follows that the instantaneous deceleration a_d of the vehicle is given by

$$a_d = \dot{v}_{sc} = \frac{F_d}{m_{sc}} = \frac{q_\infty}{\beta} \tag{3}$$

The deceleration of a vehicle is therefore directly proportional to the free stream atmospheric density and the reciprocal ballistic coefficient.

If we assume a box-shaped spacecraft with the volume V_{SC} and a length l_{SC}, equation (3) may be rewritten was

$$a_d = \frac{\rho_\infty}{\rho_{SC}} \cdot \frac{c_d v_\infty^2}{2 \cdot l_{SC}} \tag{4}$$

From (4) follows that for a given free stream velocity, the deceleration is directly proportional to the density relation between the ambient atmosphere and the spacecraft. The deceleration profile can therefore be placed at any desirable altitude range of a given planetary atmosphere, so long as the necessary ballistic coefficient is technically feasible.

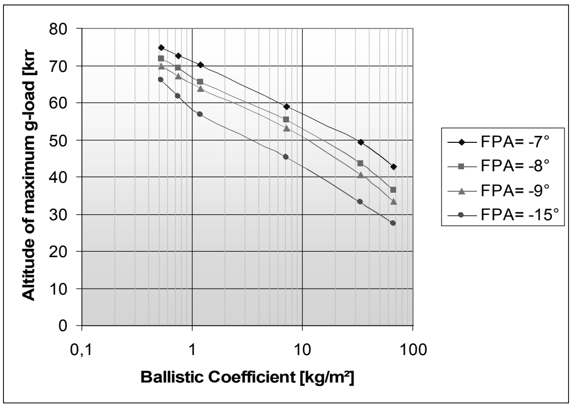


Fig. 3: Study of the altitude of the maximum deceleration point with respect to the ballistic coefficient for various Flight Path Angles (FPA) at entry into the Martian atmosphere.

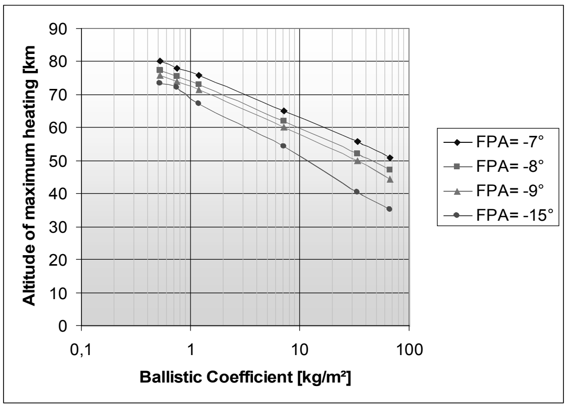


Fig. 4: Study of the altitude of the maximum heating point with respect to the ballistic coefficient for various Flight Path Angles (FPA) at entry into the Martian atmosphere.

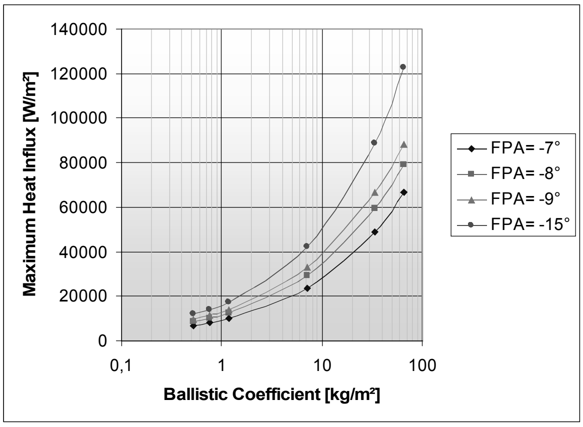


Fig. 5: Estimate of the maximum heat flux with respect to the ballistic coefficient using the heat flux model of Sutton & Graves [12] assuming a sphere of 5m radius.

Unfortunately no practical analytical method exists to calculate atmospheric entry trajectories in greater precision except for very complex ones,, such as [13], which are cumbersome to apply [14]. This makes numerical integration the method of choice. To exemplify the effects discussed above and to investigate possible mission scenarios, entries into the Martian atmosphere were studied with a numerical code using the European Mars Climate Database (EMCD) [15], assuming a perfectly spherical spacecraft entering the atmosphere at a velocity of 4.5 km/s. The trajectory points of maximum deceleration and heating are shown in Fig. 3 and Fig. 4 as a function of the ballistic coefficient and the flight path angle (FPA) at entry.

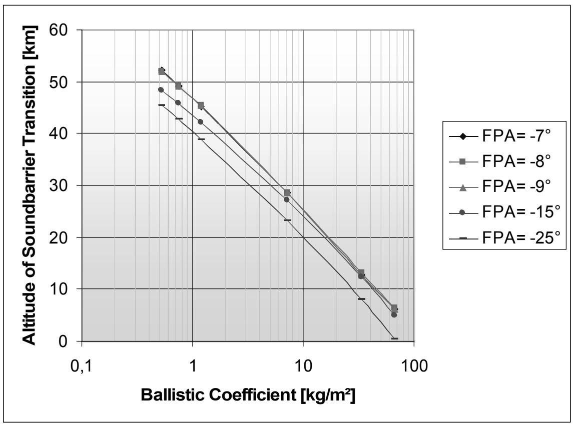


Fig. 6: Study of the sound barrier transition altitude for a vehicle landing on Mars with respect to its ballistic coefficient for various entry angles.

The maximum heating rate per unit area was calculated based on the model of Sutton and Graves [12] and is given in Fig. 5. The total descent time is given in Fig. 7 and the sound barrier transition altitude (SBTA), in Fig. 6.

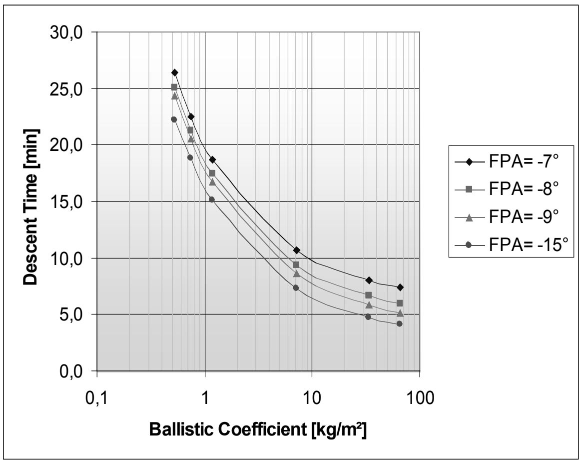


Fig. 7: Variation of descent time with ballistic coefficient for several entry angles (time from atmospheric boundary to mean radius of Mars).

As far as heating is concerned, it can be shown that for entry bodies with high ballistic coefficients the aerodynamic compression wave heating is transferred to the spacecraft wall mainly through convection and that compression wave radiation is a minor contributor [16]. There is no evidence to believe that this important relationship is not true for low ballistic coefficients, although it should be noted that no flight data of such vehicles exist. Our CFD analyses however show that compression waves of vehicles with low ballistic coefficients exhibit no differences to conventional entries that would suggest a more opaque compression wave, or one that radiates more intensely.

It should be further noted that blunt entry bodies heat up considerably less than sharper, pointy objects. However, blunter objects exhibit less directional stability than sharper objects. This makes the design of an atmospheric entry vehicle with certain stability requirements (such as crewed capsules) an exercise in trade between thermal protection system materials and active attitude control system requirements .

On a probe intended for high altitude atmospheric sounding however, landing precision and directional stability may be of lesser concern.

2.2 Ballute Spacecraft Configuration

We take the liberty to define as a ballute spacecraft every space vehicle intended to enter a planetary atmosphere with a ballute, whereby the term ballute refers to any inflatable object designed to significantly influence the vehicles aerodynamic properties. We then divide it into two main parts: the instrument pod (or payload pod) and the ballute itself.

The instrument pod is the part which carries the payload and other actively functional spacecraft elements, such as electrical and data handling subsystems and all required support structures. The pod is therefore a small, dense, and hard object, with a structure made primarily of metals or rigid composites.

The ballute on the other hand is a large and light object, gossamer as compared to the pod, with a thin and most likely flexible skin made from some fabric or film or a laminate thereof. These definitions are used regardless of spacecraft configuration and shape. The ballute can either be towed by the pod (through a harness) or clamped to the pod, depending on design drivers.

In terms of engineering, targeting the lowest possible ballistic coefficient is a quest for the lightest and biggest craft which is technically practical. If a mass to volume ratio (or overall spacecraft density) can be found which is on the same order of magnitude as the atmosphere, then static lift becomes a significant contributor to increasing scientific measurement time. If so desired and technically possible, a spacecraft may be designed that does not descent to the ground, but instead lingers at its equivalent density altitude like a super-pressure balloon, providing prolonged floatation.

The emphasis on a low ballistic coefficient makes a spherical ballute the most favourable option, as it has the best surface to volume ratio and a reasonably high coefficient of drag.

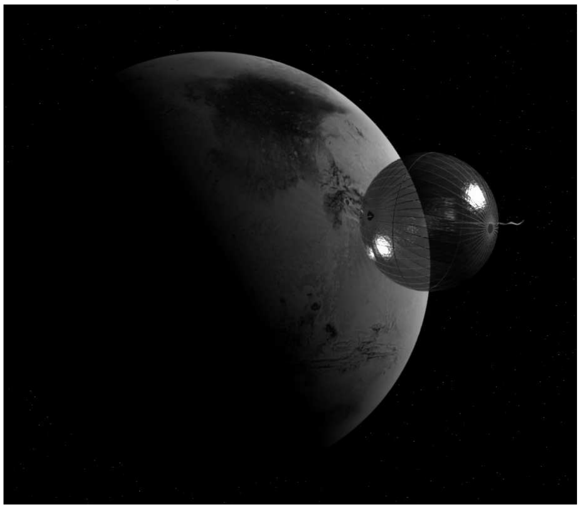


Fig. 8: Artist's impression of the ARCHIMEDES ballute, approaching Mars [17].

Roll and tumbling may be desired by some instruments. On the Huygens probe that parachuted to the surface of Saturn's moon Titan, a roll rate was artificially introduced by spinning vanes. On a spherical ballute however, asymmetric vortex shedding provides for a sufficient roll rate without the need of additional aerodynamic measures. Since landing precision is of no concern, a purely spherical ballute appears to be the best option. No burble fence or strings need to be used to control vortex shedding for the prevailing case, but should be kept in mind as a design option in case roll rate control becomes a requirement for some instruments.

The design suggested herein uses an over-pressure type spherical balloon as an inflatable hypersonic drag device with the instrument pod clamped to one of its poles (see Fig. 8). Through its over-pressure design, it combines the aerodynamic drag of a parachute with the static lift of a Helium balloon. The configuration remains unchanged throughout the entire entry, descent an landing (EDL) phase, thus reducing the risk of mechanism failure to a minimum.

2.3 Mission Design Options

Four principal mission design options are available: The first is a descent to a floatation altitude, the second a descent to the ground, and the third and fourth the same as before except that they employ an aerobreaking phase with several passes before their final entry.

In this paper, we will focus on descending to the surface of Mars with a ballute. This is desirable if scientific mission requirements call for a complete altitude coverage right down to the ground, or more simply if they do not specifically require prolonged floatation. Alternatively, if we consider the ballute as a method of delivering payloads to the surface, it might come in handy that the high-altitude deceleration profile facilitates landing at sites of higher elevation.

As far as structural engineering and practical ballute building is concerned, the advantage of this concept is the fact that the ballute body can be heavier and smaller as in a design which is required to provide for extended floatation, allowing the use of readily available skin materials and achievable seaming requirements to arrive at a lower yet more realistic mass estimate. Note also that any design intended to float for more than half a day would require an instrument pod with a thermal design that survives a Martian night. For any small payload, this is very hard and usually requires a radioactive heating unit (RHU).

In addition to this concept, the ballute may plough through the outer atmosphere layers repeatedly before loosing enough energy to finally descend deeper into the atmosphere. The process is referred to as “aerobreaking”. The advantage of this method is that more measurement time becomes available in the outer atmosphere and deceleration loads are lower. However, the entry corridor is much narrower. As a result, navigational requirements are much higher and the knowledge of prevailing atmospheric conditions has to be much more precise.

The steep entry limit is obviously an entry angle leading to just two atmospheric passes, or a scientific minimum requirement. The shallow entry limit is given by available battery power and ballute envelope durability, as each atmospheric pass will subject the ballute hull to the highly abrasive hypersonic flow field as well as mechanical and thermal load cycles.

3 FLIGHT DYNAMICS ANALYSIS

3.1 Trajectory and Mission Design

All analyses were based on 3D trajectory simulations [18], including steady state [19] and transient [20] thermal analyses. Critical points along the hypersonic flight trajectory were further studied in in detail using CFD analyses [21][22]. Eventually, aeroelastic effects were investigated by performing modal and deformation analyses [23][24].

3D trajectory simulation were made with a numerical code based on the powerful radio science mission simulator (RSS) [25] developed at the Institute of Space Technology of the University of the federal Armed Forces of Germany (UBW) [26]. Originally conceived to simulate interplanetary missions with an emphasis on preparing and analysing radio sounding experiments, the RSS is a highly versatile and very precise tool for numerical trajectory analysis. It already provides a full orbit integration including perturbation forces as subtle as solar radiation pressure, the gravity of other celestial bodies and a host of other minuscule forces. It also comes with a gravity potential model of Mars of the highest known degree and the Mars Climate Database as a reference atmosphere [15]. In addition to this, the location of planets and any point on their surface can be calculated. Another great advantage of the RSS is that the calculations of vehicle speeds and locations were routinely proven to be in excellent agreement with real world radio sounding observations done with the space probes Mars Express, Venus Express and Rosetta.

For atmospheric entry trajectory simulations with the RSS, atmospheric drag was added to the set of perturbation forces [18]. Aerodynamic heating was estimated in all trajectory calculations using the simplified analytical method of Sutton & Graves [12].

The variation of the drag coefficient with Mach number was directly taken from wind tunnel testing results obtained by the Goodyear Corporation in the 1960s [27]. Even though the shape of the studied ballute system differs somewhat from the perfectly spherical ballute suggested herein for hypersonic entry, the effect of this difference on the Mach-number dependency of the drag coefficient is considered small as compared to other uncertainties. Especially as reference flight data for such a configuration does not presently exist anyway.

The orientation of the spacecraft was assumed to be always pod-forward, as the instrument pod pulls the centre of gravity to a location outside the geometric centre and therefore should result in a pod-forward attitude during atmospheric entry. For the simplicity of the model, the same attitude was assumed for trajectory parts outside the atmosphere. Since the deformation of the ballute during hypersonic flight is expected to be minimal, the departure of the ballute from a perfectly spherical shape was not modelled. Hence, aerodynamic lift was also not modelled.

Selected plots of a typical result are shown in Fig. 9 (in this case for an ARCHIMEDES orbit version designated Alpha-1 Norm with two nominal atmospheric entries). The analysis starts with the de-

Because of its concise nature, the RSS is also ideally suited to study the sensitivity of ballute missions and to analyse mission geometries. For the ARCHIMEDES ballute spacecraft, it is necessary to ensure that atmospheric passes do not occur during eclipse and that they are visible from the telecommunications orbiter to ensure a permanent telemetry stream during that critical mission phase.

To do this, a tool was added to the RSS that can calculate the range and visibility between two spacecraft and perform a radio link analysis [28]. Range, visibility, possible data rate, lighting conditions and their visibility from Earth of two spacecraft can be calculated and plotted. For project ARCHIMEDES, orbits of a possible telecommunication satellite and the ARCHIMEDES ballute were analysed that way, starting from the point of ARCHIMEDES' de-orbit.

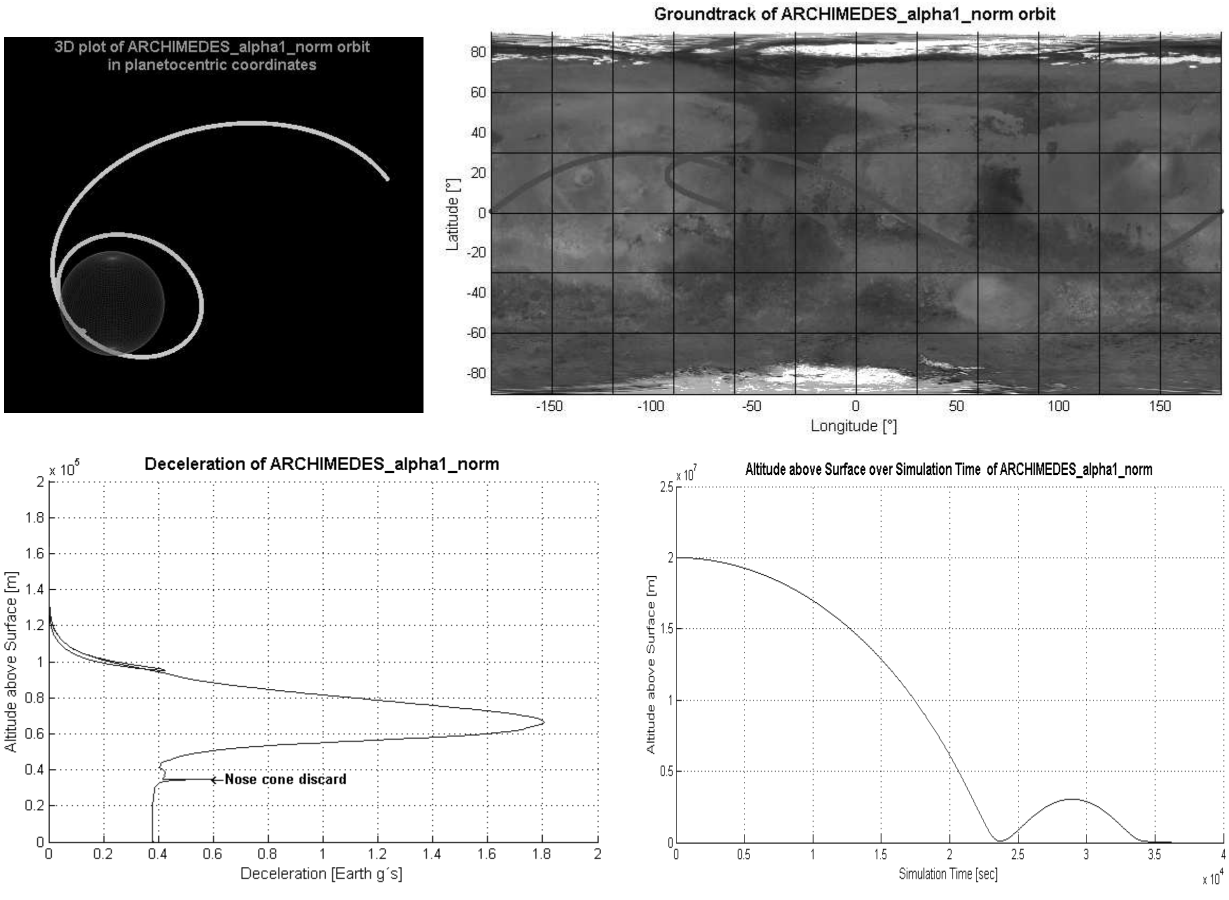


Fig. 9: Selected results of a typical ARCHIMEDES trajectory analysis (in this case the Alpha-1 Norm orbit).

orbit manoeuvre. The 3D orbit plot gives an overall picture of the orbit in planetocentric coordinates and the ground track plot shows the trajectory ground track of the ballute across the surface. The deceleration plot clearly shows the first and second atmospheric entry deceleration peaks and the point at which the instrument pod's protective nose cover assembly is discarded.

3.2 Aerothermodynamics and Aeroelasticity

Detailed aerothermodynamic analyses were performed to improve the knowledge about the thermal and mechanical loads and the dynamics of the balloon.

Two reference points were selected for the analyses: the point of the trajectory where the maximum stagnation point temperature occurs and the trajectory point of maximum deceleration.

For Mars entry, the analysis [21] was based on a modification of the hypersonic CFD code CEVCATS-N for a rarefied high-altitude hypersonic flow field with chemical non-equilibrium [29]. CEVCATS-N is a Navier-Stokes solver, originally developed for low-speed continuum flows by the DLR Institute of Aerodynamics and Flow Control. It is based on a structured mesh storing flow variables at cell vertices. Adapting this code for hypersonic entry requires the important assumptions that the shock front is transparent to thermal radiation, which has already been confirmed for a continuum flow and entry velocities much higher than for the prevailing case [16].

Despite the high altitude of the deceleration profile and the low density of the Martian atmosphere there, the application of a Navier-Stokes solver is justifiable because of the big diameter of the ballute.

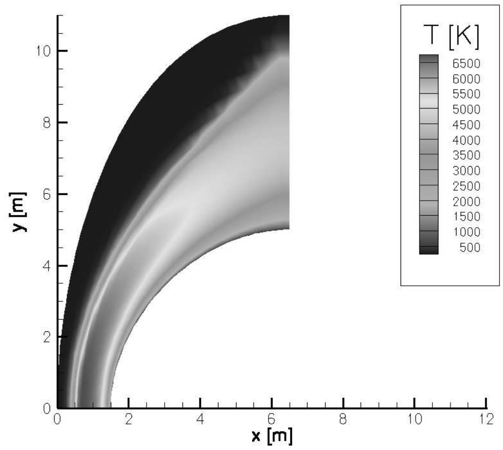


Fig. 10: CFD results for the flow-field temperature of an ARCHIMEDES type ballute spacecraft at the point of maximum heating rate as determined with CEVCATS-N [21].

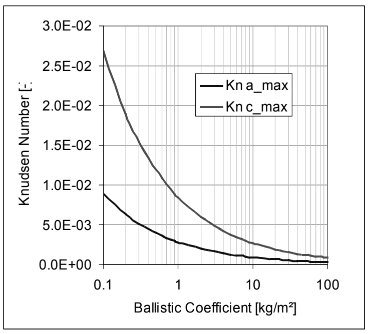


Fig. 11: The Knudsen number for the trajectory points of maximum heating rate and maximum deceleration as a function of the ballistic coefficient for 36 kg spherical ballute spacecraft entering the Martian atmosphere at -10°.

Fig. 11 shows the Knudsen number for the points of maximum heating and maximum deceleration of ballute spacecraft like the ones studied herein, entering the Martian atmosphere at -10°. The graph shows that all practical ballutes with realistic ballistic coefficients can be regarded as being in a continuum flow during these two critical mission phases and can therefore be treated with Navier-Stokes solvers for CFD analysis.

Dust effects within the flow were not considered, as we may conclude from limb sounding experiments with the HRSC camera on Mars Express [30] that dust particles in the atmosphere do not reach altitudes at which the main deceleration profile of a configuration such as ARCHIMEDES occurs.

Results obtained with CEVCATS-N for a typical ARCHIMEDES Alpha-1 Norm type entry trajectory as calculated with the RSS and a simple atmosphere model composed of a 97% CO_2 and 3% N_2 are given in Fig. 10, where flow field temperatures are shown for the point of maximum heating. The maximum flow field temperature directly at the wall of the ballute was determined to be 470K.

Two different finite element (FEM) analyses using MSC Nastran were made for a 10-m ballute. Both were used to analyse static deformation at the two critical trajectory points of maximum heating and maximum deceleration. The first model was a simple sphere with no detail whatsoever [23], that provided a quick look and an estimate of the general ballute envelope behaviour. Based on experience gained with this model, a more sophisticated model including seams and polar caps was made for an in-depth analysis [24]. The inertia relief method was used for this, which allows the simulation of unconstrained structures (such as flying vehicles) in a static analysis. Instead of fixing it at dedicated points, the structure is constrained by its own moment of inertia.

For the modal analysis, only a simple sphere was modelled, however including the filling gas. Nastran's Normal Mode analysis was used, which neglects all attenuation effects. The first eigenmode for a 10-metre ballute was found around 41Hz and the second around 50Hz. Increasing the envelope skin thickness by a factor of 6 decreased the first eigenfrequency to 39Hz and the second to 47Hz. The skin thickness is therefore found to be negligible and reinforcements and seams can safely be omitted in such an analysis. Reducing the diameter however does have a significant effect. A ballute of 5 metres diameter was found to have its first natural eigenfrequency at 80Hz, and the second at 96Hz.

3.3 Thermal Analysis

To analyse the thermal behaviour of the ballute during all mission phases (in space, during hypersonic flight and final descent), a simple yet effective thermal node-model based on [20] was created and included in the numerical mission evaluation computation. Heat sources are solar and planetary radiation, latent heat from the instrument pod and hypersonic compression wave and friction heating. Heat sinks are the space environment and the planetary atmosphere.

Results from this thermal model can be used as a basis for the material analysis. The above calculations show a temperature which is a little bit lower than the temperatures obtained from CFD analyses, since material heat capacities and balloon internal convection were taken into account. However, the general heat-flux distribution obtained from the CFD analyses were used in the implementation of this thermal model, which was made part of the RSS trajectory simulation.

Typical results, plotted against mission time, are shown in Fig. 12. In this set of plots, all nodes are shown simultaneously. Note the short eclipse period during which the temperature drops as low as 150 K for all nodes, requiring active heating for the payload and avionics suite.

Another important fact is that the comparatively low skin temperatures during entry (as compared to classic probes) are also possible because of heat being radiated off inside the ballute from the “hot” hemisphere towards the “cool” hemisphere in the vehicle's wake field.

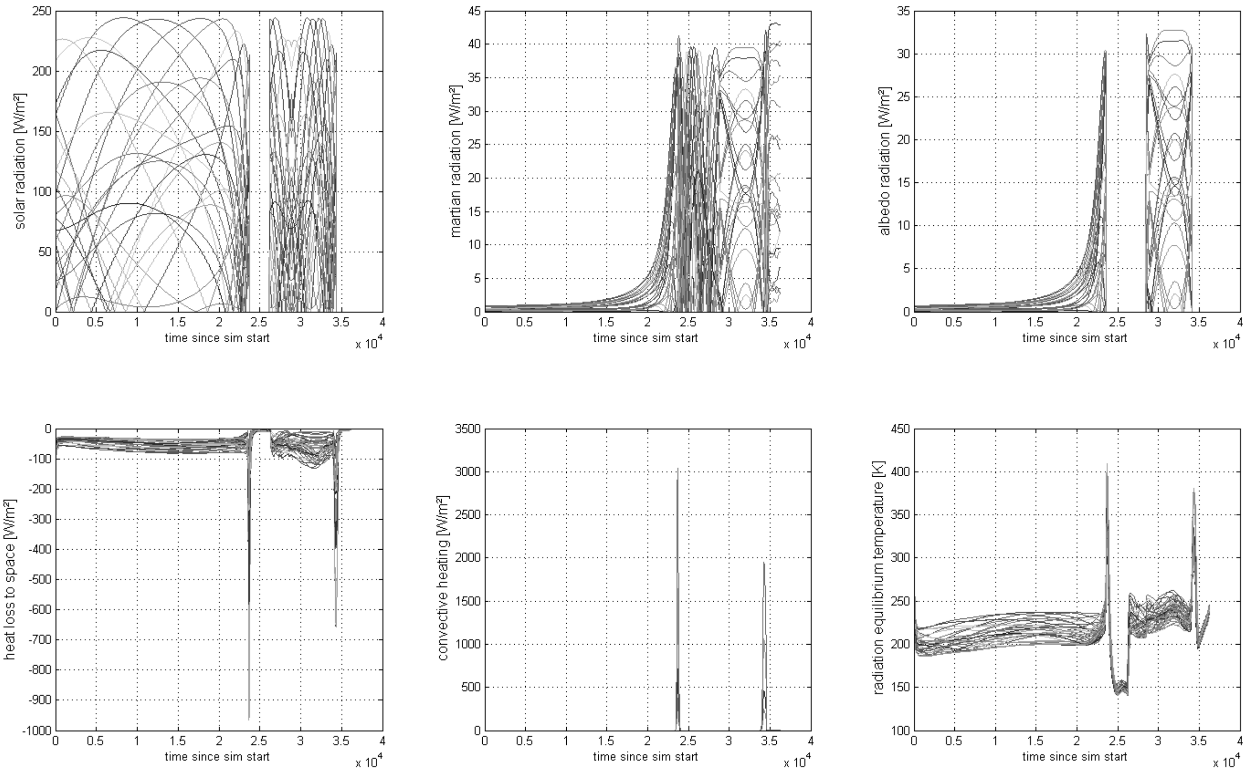


Fig. 12: Typical heating results for all vertices across the ballute surface for an ARCHIMEDES Alpha-1 Norm type entry trajectory.

4 MATERIAL SELECTION AND ANALYSIS

Finding the right material for the ballute envelope is of paramount importance and represents a key technology. In order for the ballute to resist high pressure and temperature differences during hypersonic flight, yet be light enough for a reasonable mission, it has to be made of a high performance thin-film material or a laminate thereof. Such materials were tested under expected mission conditions at the Institute of Material Science at the UBW. Together with the MSD, seaming and production techniques were also explored, including the development and testing of high strength high temperature resistant adhesive tapes by Lohmann Tapes of Neuwied, Germany and a number of welding techniques.

The ballute envelope has to be flexible enough to be packed into an acceptable volume, stable enough to retain its mechanical properties while packed and light enough to fulfil mission requirements. It has to be impermeable to gas and strong enough to hold up during hypersonic flight. Note that this last point not only includes high temperatures, but also chemical and mechanical stability in a hypersonic flow field environment, which might contain chemically aggressive constituents dissolved in the shock wave. Last but not least, the material has to conform to space qualification standards, as defined in ECSS-Q-70-71A rev. 1 [31] and ECSS-Q-70-02A [32], as well as planetary protection requirements [33]. The material also has be readily available in large enough quantities to fashion a series of ballutes from it.

An initial survey of available materials and their suitability for a Mars balloon mission was made for the ARCHIMEDES project by the Institute of Statics and Dynamics of Aerospace Structures (ISD) at the University of Stuttgart, an effort that later even received ESA funding [34]. The study concluded by identifying several possible plastic thin films for surface or air deployed balloons, but found that most materials fail at conditions encountered during hypersonic flight. However, studies of balloons for the lower Venusian atmosphere exist, which is also a high temperature and chemically aggressive environment [35][36].

Based on all of these finding, two polymers named UPILEX®-S and -R have been identified as the most promising candidates. The thermoplastic material UPILEX® is a polyimide which is known for good mechanical properties at high temperatures. Another material, known as Polyphenylbenzobisoxazole (PBO) promises much better performance, but is not commercially available as a thin film and appears to be sensitive to damage induced by radiation. The company Foster-Miller of Waltham, Massachusetts, has already built subscale test balloons made of PBO for NASA [37] and was willing to deliver small film samples for testing, so the applicability of PBO for ARCHIMEDES was also investigated.

Results show that even much degraded PBO offers a far superior performance than UPILEX®, but availability remains a huge problem. The UPILEX® subtype SN also comfortably exceeds mission requirements, but is almost impossible to pack, because it is stiff and easily breaks. UPILEX®-RN, in turn, boasts excellent processing and packaging properties, but mission margins are minimal.

It is therefore practical to either investigate the necessary technology to manufacture large PBO thin films in large quantities or to design a ballute with UPILEX®-RN, that has structural reinforcements made of PBO where appropriate.

5 FLIGHT TESTS

5.1 Parabolic Flight and Space Deployment Tests

To test the TDS for functionality and obtaining data with which to correlate mathematical predictions of the balloon deployment speed, a test system was successfully flown in June 2005 on ESA's 40th parabolic flight campaign in Bordeaux. This allowed simulating various failure modes and testing the packaging method for its unfolding characteristics [38][39]. Experience and insight gained during this campaign also allowed continuing the development of hardware for space tests.

To observe the behaviour of the deployment system design and the deployment of a dummy ballute in space, a test was planned named REGINA and was completed in just 8 months. It was flown on the REXUS-3 sounding rocket on April 5, 2006. To observe the deployment, a dedicated Camera Module inside the rocket interface ring was used as an observation platform. A single stage Orion solid-fuel rocket took the REGINA flight system to the edge of space at 93 km, on a ballistic trajectory. The rocket was sponsored and operated by the MoRaBa group of the DLR space operations centre in Oberpfaffenhofen and the Swedish Space Corporation.

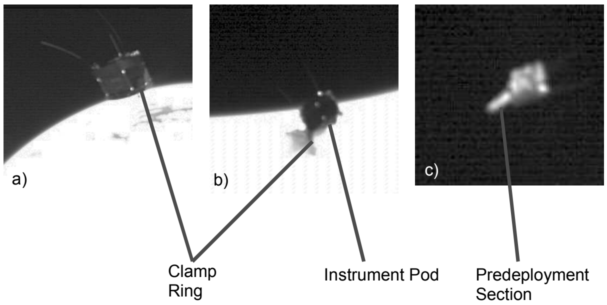


Fig. 13: Deployment sequence of REGINA in space, as it drifts away from the rocket's payload section: a) before deployment; b) clamp ring rotating away from the system (blurred); c) pre-deployment section extending away from the blossom leafs (Images: Fleischmann / TU München).

The instrument pod carried no instruments and only acted as a mass dummy. Videos were recorded by 10 solid state video camcorders (VCC) with VGA resolution. Images show Images obtained by the Camera Module and the T-REX fish-eye camera experiment, built by the Technical University of Munich, showed that the system deployed the ballute dummy correctly during the flight (Fig. 13). No significant inflation under residual gas could be observed. The results were deemed good enough to decide in favour of a continuation of the flight testing programme.

5.2 The MIRIAM Spaceflight Test

For the evaluation of the entire ballute technology, a representative test mission was designed. The name of this test mission is MIRIAM, forming an acronym for: "Main Inflated Reentry Into the Atmosphere Mission test (for ARCHIMEDES)". For the REXUS 4 ballistic rocket campaign in October 2008 a MIRIAM experiment was designed and flown to test the release, inflation, separation and atmospheric entry of a hypersonic drag balloon. The space flight system composite consists of 3 major elements:

1. The Miriam ballute spacecraft which comprised an instrument pod and the helium-filled hypersonic drag balloon of 4m diameter and a total vehicle mass of around 5.6 kg .
2. The Service Module (SM) which contained the inflation system, structural box, release mechanism, SM telemetry and a live television subsystem. It also contained a set of cold gas thrusters. These thrusters pulled the Service Module away from Miriam after inflation.
3. The Camera Module (CM) which stayed on the rocket. It provided a platform for a camera subsystem and also provided the main release mechanism and structural interface for the Service Module / Miriam composite.

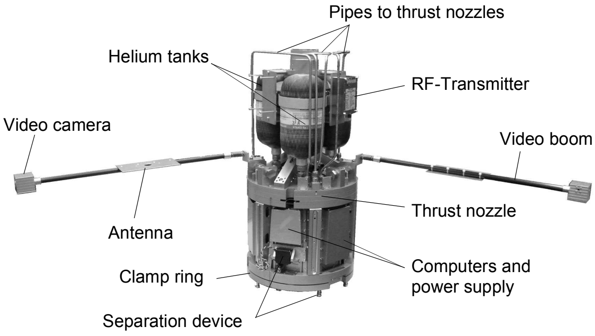


Fig. 14: The MIRIAM Service Module structure (actual flight model shown).

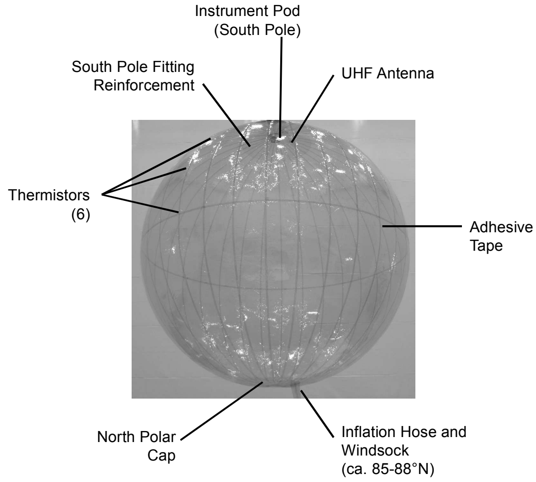


Fig. 15: The MIRIAM flight model ballute spacecraft during a pressurization test at IABG Ottobrunn.

All three elements combined formed the MIRIAM Flight System Stack. It was mounted underneath the rocket's nose cone assembly, which had to be jettisoned to allow separation of the spacecraft.

The actual flight systemhad a grand total of 39kg and stood over 1m tall. The overall diameter was 14 inches (355mm), a standard sounding rocket diameter. Fig. 14 shows a picture of the service module structure. The entire flight system stack was tested for launch vibrations and vacuum deployment at IABG Ottobrunn.

MRIAM's pod was instrumented for flight analysis purposes. On the actual vehicle, an FMI-Helsinki-provided ATMOS-B pressure sensor was installed behind the rear pod bulkhead, facing the ballute interior. The magnetometer for MIRIAM (MiriMag) was contributed by the IGEP institute and the company MAGSON of Berlin. Paired with an optical still image camera, it was to yield attitude information. The still image camera was a commercially available low resolution unit which could be integrated cheaply and easily and was sufficient for an occasional attitude fix, in combination with the other sensors. A suite of two different sets of accelerometers built by the ARCHIMEDES team and the universities in Iasi and Pitesti, Romania were to give deceleration and roll rate information.

The Service Module telemetry and TV signals during the flight of MIRIAM were recorded, until the signal was lost (see also Fig. 16 and Fig. 17).

In addition to transmitting live telemetry, the Miriam Instrument Pod and Service Module flight controllers were equipped with internal permanent solid state memories, to record certain flight parameters at higher resolution. To be able to retrieve the systems, both the Miriam Instrument Pod and Service Module were equipped with radio beacons.

Because MIRIAM is supposed to resemble a valid model of ARCHIMEDES, the ballute spacecraft was designed such that its aerodynamic and flight dynamic properties resembled the ones planned for ARCHIMEDES as good as possible. The scale of the model is defined by the dimension relation of the ballute.

Unfortunately, MIRIAM's main interlock bolt number one jammed upon release, a behaviour that was not observed during pre-flight tests.

As a result, the MIRIAM Service Module stayed attached to the rocket far longer than planned and separation occurred but the separation velocity was much slower than expected. Subsequently this led to a an improper deployment of the ballute. Pictures of the

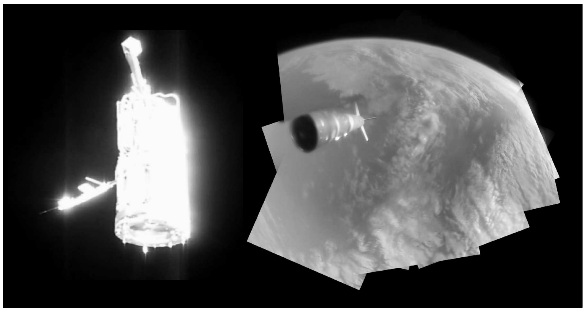


Fig. 16: The MIRIAM Service Module, as seen from the rocket payload section (left) and the payload section, as seen from the service module (right).

service module and the rocket payload section can be seen drifting away from each other in Fig. 16.
Therefore the balloon was set free with little over 10% of its intended amount of filling gas. A picture of the ballute in space is Fig. 17.
All other subsystems functioned nominally and behaved as expected from tests, leading to a partial success of the mission. However, a repetition of the flight test must be made and a second MIRIAM test, MIRIAM-2, is currently under study for a possible 2013 launch.
Primary technical data of MIRIAM and ARCHIMEDES is givenin Table 1.

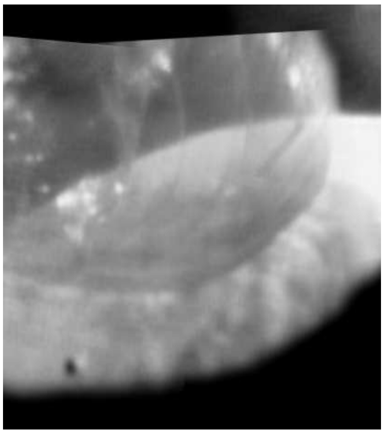


Fig. 17: The MIRIAM Ballute Spacecraft during deployment.

Table 1: MIRIAM technical data compared to ARCHIMEDES.

Value	ARCHIM	MIRI	Ratio M:A
Mass Instr. Pod [kg]	ca. 18.3	2,9	1:6.30
Mass Ballute [kg]	ca. 15.1	2,6	1:5.80
Mass Ballute S/C [kg]	ca. 33.2	5,4	1:6.10
Pod : Ballute [-]	ca. 1.21	1,12	1:1.08
Ballist. Coeff. [kg / m²]	ca. 0.54	0,59	1:0.92
Ballute Diameter [m]	ca. 10	ca. 4	1:2.50

6 THE ARCHIMEDES MARS BALLOON PROBE

ARCHIMEDES is the actual hypersonic drag ballute concept designed to probe the atmosphere of planet Mars as outlined in the scientific concept [1]. The project is currently under active development by the Mars Society Germany and the Universität der Bundeswehr München and also supported by several other research institutions and industrial companies.

6.1 Spacecarft Overview

Aside from the scientific scope of project ARCHIMEDES, an important goal of the project is to demonstrate and qualify the ballute technology for entry into planetary atmospheres at high velocities on a representative mission.
The suggested payload suite is reflected in the mission name, which forms an abbreviation for "Areal reconnaissance Robot Carrying High-resolution Imaging, a Magnetometer Experiment and Direct Environmental Sensors". Hence, the primary payload constitutes a high resolution camera suggested by the DLR centre for planetary exploration Berlin, a magnetometer experiment studied jointly by the IGEP institute of the technical university of Braunschweig and the private company MAGSON of Berlin and the so called ATMOS-B weather sensor suite by the Finnish Meteorological Institute (FMI) of Helsinki. Additionally a pyrolytic compression wave temperature experiment by the IRS institute of the Technical University of Stuttgart and a high sensitivity accelerometer built jointly by the Technical Universities of Iasi and Pitesti in Romania are foreseen to ride in the nose cover assembly, which will be jettisoned after the transition to subsonic speeds.
The reference mission has three major flight system elements: the AMSAT P5-A Mars orbiting satellite, the joint AMSAT-Archimedes Propulsion System JPS and the ballute system ARCHIMEDES (see Fig. 18), although the concept would work for any Mars satellite and does not solely rely on an AMSAT derived design.

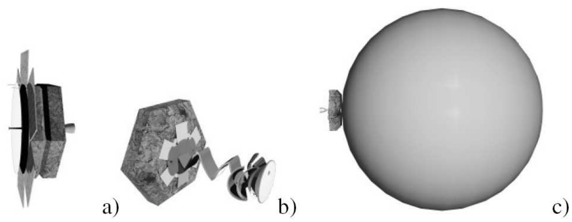


Fig. 18: a) The AMSAT P5/A in cruise configuration. b) The ARCHIMEDES vehicle being deployed from the Joint Propulsion System Module (JPS). c) ARCHIMEDES fully inflated with the JPS module still attached to the North Pole.

Total projected Earth orbit injection mass is 650kg, of which 110 kg are the orbiter. ARCHIMEDES accounts for 66 kg, including a 34 kg ballute spacecraft, plus 32 kg inflation and transportation systems. A 20% mass growth allowance is added to this budget, to account for design uncertainties, so that the total mass reserved for ARCHIMEDES in the combined mission study is 80 kg.
The ballute was designed to matches a desired ballistic coefficient of around 0.5 kg/m².

6.2 Trajectory

A reference target orbit, based on the AMSAT study, with a 4 000 km pericentre altitude and a 20 000 km apocentre altitude was chosen which is inclined by 30° against the equator. Entering the atmosphere during daylight greatly eases the thermal constraints and is a requirement for the camera experiment. Therefore, the

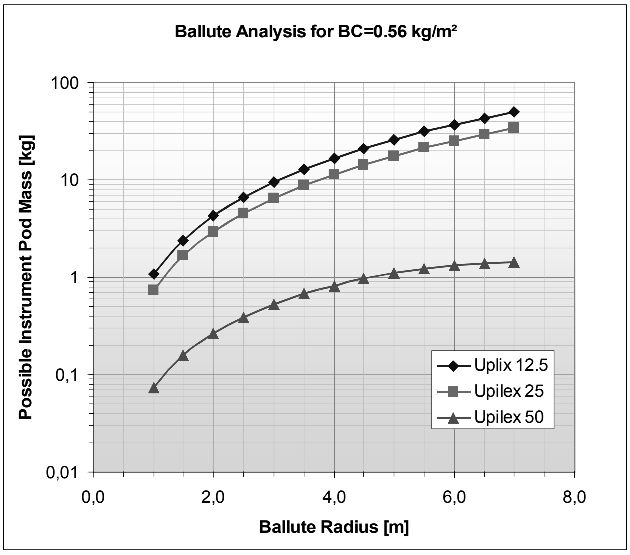
orbit has to be oriented such that its periapsis and with it the atmospheric entry point lies on the sunlit side of the planet. As a result of that, the apoapsis is eclipsed.
To get the camera heads a scanning view of the environment, the ballute will be left to rotate freely in the air stream.
To prolong measurement time in the upper atmosphere, multiple entries are planned. A small risk exists that the vehicle exits the atmosphere on a suborbital trajectory that re-enters half-way around the planet on the night side. This would end the thermal life of the instrument pod after aerodynamic heating cools off and would jeopardize the camera experiment. However, Monte-Carlo analyses of some 5000 trajectories show that such a risk is acceptable, as most missions complete enough of a full orbit to descend on the sunlit hemisphere.
To make ARCHIMEDES as light as possible, the ballute is filled with Helium, which adds significant static lift to aerodynamic drag during final descend. The resulting trajectory allows continuous measurement times of several hours ranging from the outermost atmosphere layers down to the ground.
Because a skipping entry presents a delicate navigation problem, a two-step approach strategy was designed. In this scenario, the JPS performs a total of two or three de-orbit burns. The first one to lower the periapsis of the JPS orbit to around 400km above the atmosphere interface. The orbit is then determined through radio ranging from the ground and the orbiter to calibrate the engine running on an almost empty tank system. If required, the manoeuvre can be repeated, to bring the JPS even closer to the atmosphere. The final deorbit impulse can be given either by the precisely calibrated main engine or cold-gas RCS thrusters which have a higher accuracy.
Once the final deorbit impulse has been made, the JPS coasts towards the atmosphere. About three hours out from atmospheric entry, the ballute spacecraft is deployed and inflated and the JPS eventually discarded.
The actual mission elapsed time and number of atmospheric entries have been determined by performing an error analysis and by studying “shallow”, “nominal” and “steep” cases of a “normal” (34 kg entry mass) and “heavy” (65 kg entry mass) spacecraft. In a “heavy” and “shallow” case, the given spacecraft will enter the atmosphere up to 8 times before final descent, while in a “steep” case the system will decelerate directly onto its decent trajectory, without any further orbit revolution. The nominal case is a 2-entry mission.

7 OTHER APPLICATIONS

Many space applications would greatly benefit from the inflatable hypersonic drag device known as “ballute”. Aside from its use as a scientific probe, it might also be used as an entry vehicle to deliver a payload to the surface of a planet that is inaccessible to conventional parachutes or requires a heat shield too big for available rocket fairing diameters. Planetary aerocapture and aerobreaking applications could also make good use of a ballute, as would an orbital recovery system for astronauts. Tourists into extreme sports, who have done everything in their life except a hypersonic entry from space, might find it worth the expense to fly a ballute from a suborbital trajectory to the ground. Eventually, it is even conceivable to build a solar sail ballute.

8 CONCLUSION AND OUTLOOK

The study shows that existing balloon technology, as well as existing inflation systems technology, can be combined to create a robust inflatable drag body (referred to as “ballute”) that is able to withstand the mechanical and heating loads during atmospheric entry. It could also be shown that a high performance ballute may be used to achieve a certain scientific goal by reaching altitudes otherwise inaccessible.
An underlying theory and a design guide for such a vehicle can be used for the study of a ballute spacecraft in general, regardless of purpose. It was used herein for the design of the suggested Mars mission ARCHIMEDES and its flight test vehicle MIRIAM.
The theory shows that with available film gauges, a viable scientific mission can be flown with sufficient ballute payload mass for a meaningful scientific instrument package (see Fig. 19).
Material analysis showed that thin films with these gauges exist that are sufficiently strong, but that procurement might be an issue with the most suitable candidate material (PBO thin film).
How well the theory works in practice could not be verified. The flight test system MIRIAM never got deployed properly after a bolt malfunctioned in space. As this test is a necessary confirmation of the technical concept put forward herein, it must be repeated.
It could already be shown, however, that a ballute for space applications can be developed and that it is a very real as well as promising alternative technology option for atmospheric entry.



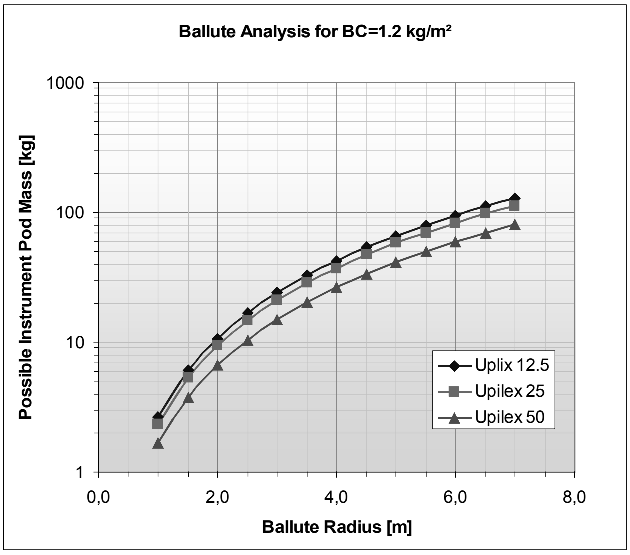


Fig. 19: Possible Instrument Pod mass as a function of ballute radius for various UPILEX films and desired ballistic coefficients (without considering material strength).

9 REFERENCES

[1] Grieger,B., et al., "*The ARCHIMEDES Mission Science Proposal*", Mars Society Deutschland, Max Planck Institut für Aeronomie Lindau, 2005

[2] H.S.Griebel, "*Methoden der Ballonentfaltung auf dem Mars Konzepte zur praktischen Durchführungund Versuchsanordnungen zu deren Erprobung RT-DA 03/01*", Technical University of Munich, 2002

[3] Jacques Blamont, "*Planetary Balloons*", Springer, 2008

[4] Raymond Turk, "*Pressure Measurements on Rigid Model of Ballute Decelerator at Mach Numbers from 0.56 to 1.96*", NASA, 1966

[5] Theo Knacke, *"Parachute Recovery Systems Design Manual"*, Para Publishing, 1992

[6] D.Wilde, S.Walther, K.Pitchadze wt. al., "*Inflatable Reentry and Descent Technology (IRDT) – Further Developments*", 2 nd International Symposium of Atmospheric Reentry Vehicles and SystemsArcachon, France, 2001

[7] Angus McRonals, "*A light Weight Inflatable Hypersonic Drag Device for Planetary Entry*", JPL, 1999

[8] Jeff Hall, "*A Review of Ballute Technology for Planetary Aerocapture*", NASA, JPL, 2000

[9] Peter Gnoffo, Brian Anderson, "*Computational Analysis of Towed Ballute Interactions*", AIAA, 2002

[10] Frank J. Regan, Satya M. Anandakrishnan, "Dynamics of Atmospheric Re-Entry", AIAA, 1993

[11] P.H.Zipfel, *"Modeling and Simulation of Aerospace Vehicle Dynamics"*, AIAA, 2000

[12] K.Sutton, R.A.Graves jr., "*A General Stagnation-Point Convective-Heating Equation for Arbitrary Gas Mixtures*", NASA, 1971

[13] N.X.Vinh, J.R.Johanessen, J.M.Longuski, J.M.Hanson, "*Second-Order Analytic Solutions for Aerocapture and Ballistic Fly-Through Trajectories*", Journal of the Astronautical Sciences, 1984

[14] Kristin Gates Medlock, James M. Longuski, "*An Approach to Sizing a Dual-Use Ballute for Aerocapture, Descent, and Landing*", 2006

[15] Fourget, Hourdin, Fournier, Hourdin, Talagrand, Collins, Lewis, Read, Huot, "*Improved general circulation models of the Martian atmosphere from the surface to above 80 km*", 1999

[16] R.A.Micheltree, M. DiFulvio, T.J. Horvath, R.D.Braun, "*Aerothermal Heating Predictions for Mars Microprobe*", NASA, 1998

[17] Michael Schwarze, "*Erstellen einer 3D-Animation für die geplante Marsmission ARCHIMEDES*", Institut für Raumfahrttechnik, UniBw München, 2009

[18] M.Zuban, "*Untersuchung des Atmosphäreneinflusses auf Raumflugbahnen im Radio Science Simulator des Instituts für Raumfahrttechnik*", UniBw München, 2005

[19] C. Fiebig, "*Erstellen eines ersten Thermalmodells f ̈ur das Mars BallonExperiment Archimedes im Weltraum*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2005

[20] L.Kotte, "*Thermalanalyse der Mars Erkundungssonde ARCHIMEDES*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2007

[21] T.Schramm, "*Untersuchung der aerothermodynamischen undmechanischen Wechselwirkung amMars-Ballon ARCHIMEDES*",

Institut für Thermodynamik, Universität der Bundeswehr München, 2007
[22] H.Pasemann, "*Implementierung von Slip-Randbedingungenin das Programm CEVCATS*", Institut für Thermodynamik, Universität der Bundeswehr München, 2004
[23] M. Zuban, "*Untersuchung des Einflusses der Hyperschallströmung auf die Form eines Ballons beim Eintritt in dieMarsatmosphäre*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2006
[24] T.Högner, "*Aeroelastische Analyse des ARCHIMEDES Ballons*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2008
[25] J.Selle, "*Planung und Simulation von Radio-Science-Experimenten interplanetarer Raumfahrt-Missionen*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2005
[26] B.Häusler, O.Zeiler, G.Billig, W.Eidel, "*Satellitensimulator für Kleinsatelliten*", Institut für Raumfahrttechnik, Universit#t der Bundeswehr München, 1996
[27] Jaremenko, I.M., , "*BALLUTE Characteristics in the 0.1 to 10 Mach Number Speed Regime*", Journal of Spacecraft and Rockets, 1967
[28] T.Krauledat, "*Analyse und Simulation des Missionsablaufs der Raumsonde Archimedes*", Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2006
[29] Mundt,Ch., Griebel,H., Welch,Ch., "*Studies of the atmospheric entry of vehicles with very low ballistic coefficient*", 13th AIAA/CIRA International Space Planes and Hypersonic Systems and Technologies Conference, 2005
[30] Harald Hoffmann, Personal conversation and e-mail correspondend on unpublished work, 2005
[31] European Cooperation for Space Standardization, *"Data for selection of space materialsand processes"*, ECSS Secretariat, ESA-ESTEC, 2004
[32] European Cooperation for Space Standardization, *"Thermal vacuum outgassing test for the screening of space materials"*, ECSS Secretariat, ESA-ESTEC, 2000
[33] K.Bayler, "*Archimedes Planetary Protection Specification, V002*", The Mars Society Germany (Mars Society Deutschland e.V.), 2003
[34] A. Kunze, F. Epperlein, B. Kröplin, "*MARS BALLOON ENVELOPE STUDY*", ESA Contract No. 16212/02/NL/PA, 2002
[35] Andre Yavrouian, "*High Temperature Materials for Venus Balloon Envelopes*", 1995
[36] A. Yavrouian,; G. Plett,; S. Yen,; J. Cutts,; D. Baek, " *Evaluation of Materials for Venus Aerobot Applications*", AIAA Summer Conferences , June 28-July 1, 1999
[37] L.Rubin, J.Larouco, "*High Performance Balloon Envelope Materials for Planetary Aerobots*", Foster Miller, MA, 2002
[38] O.C.Petrache, H.S.Griebel, "*Evaluation of the ARCHIMEDES Balloon Deployment in Zero Gravity during the Parabolic Flight Test Campaign VP49*", Systems for Space Applications, 23-37, Performantica, Iasi, Romania, 2006
[39] H.S.Griebel, "*ARM Test System for Parabolic Flight VP 49Technical Manual*", Mars Society Deutschalnd e.V., Institut für Raumfahrttechnik, Universität der Bundeswehr München, 2005